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Aging Systems and Sustainment Technology

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Summary

Fatigue is a failure mode in aircraft that emerged in the fifties and sixties as a significant threat to their structural integrity. Since that time, the research community has extensively studied the phenomenon and has developed the technology to describe the propagation of fatigue cracks in a structure. This paper describes an approach that, when followed, will virtually eliminate catastrophic failures from this mechanism.

Introduction

Most engineering structures, particularly metallic components, when subjected to cyclic loading have the potential for failing below their pristine strength. Such a failure is referred to as a fatigue failure. There is a progressive degradation of the strength from cracks emanating from manufacturing damage, in-service induced damage, or intrinsic defects in the material. Constant amplitude testing is used to characterize the residual strength of a structural member after it has been subjected to a specified number of loading cycles. This paper examines the approaches that researchers have used to prevent catastrophic structural failure resulting from cyclic loading.

Except for those failures resulting from exceeding the operational envelope of the aircraft, structural failures prior to the mid-forties were rare. One reason for this is that before the mid-forties, aircraft rarely accumulated sufficient flight time on their aircraft to suffer from fatigue failures. Further, the ductile materials and conservative methods used for analysis tended to preclude failures. Experience has shown that early aircraft manufactured with ductile materials and designed based on static strength only are typically safe from failure caused by fatigue for at least 1000 flight hours. It was rare for a combat aircraft in World War II, for example, to remain operational for more than 1000 hours. The demand for improvements in performance in the late forties; however, introduced new materials with high strength, but few other virtues. Further, the demand for performance improvements reduced analytical conservatism and introduced designs that were to operate at high altitudes. The design community appeared oblivious to the consequences of their actions. Even before the time of the first flight by the Wright Brothers, fatigue was a major issue in many industries. In the railway industry alone, fatigue failures of wheels caused numerous deaths. These failures seemed to make no impression on aircraft designers. The success they had stemming from the days of the Wright Brothers appeared to continue without interruption although fatigue failures in aircraft can be traced back to the late twenties.

The reality of the consequences of aging came sharply into view for the United States Air Force (USAF) on March 13, 1958 [1] when they lost two B-47 aircraft because of fatigue cracking in the wing. It was on this day that the aging aircraft research effort started for the USAF. The USAF did not specify a service life for the B-47. Consequently, they based the design on the assumption that failure from overload was the only threat to its structural integrity. This was common practice for aircraft designed in the late forties such as the B-47. Review of the then current literature on structural design provided no

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hint fatigue was a serious consideration. However, the USAF intended to maintain the aircraft in service until 1965. The technical basis for maintaining the aircraft in service for that length of time did not exist.

The 1958 failures motivated the USAF to establish the USAF Aircraft Structural Integrity Program (ASIP). The USAF designed this program for use with new weapon systems acquisition for their inventory. The program as originally conceived defined a sequence of tasks that progressively reduced the risk. These tasks, composed of analyses and tests, included all efforts for the qualification of USAF aircraft. This concept is just as valid today as it was in 1958. The approach, although sound in its concept, had a fatal defect. The original program incorporated a reliability concept called safe life to qualify the structure for the loads environment expected in operational service. The USAF determined the safe operational life from the results of a full-scale fatigue test of the structure. They conducted this test in a laboratory environment. They divided the number of successfully tested flight hours by a factor called the “scatter factor.” The scatter factor (usually in the interval from two to four) supposedly accounted for material property and fabrication variations in the population of aircraft. The trouble with this approach was that it did not preclude the use of low ductility materials operating at high stress. Unfortunately, it was at this time that aluminum companies were introducing high strength alloys in response to the insatiable desire for improved aerodynamic performance. Consequently, the “safe life” concept did not eliminate the in-service failures the USAF designed it to prevent. The “safe life” approach adopted by the USAF in 1958 proved to be ineffective in eliminating fatigue cracking as evidenced by the failures in operational aircraft.

Probably the most significant in-service event since 1958 that changed the original version of the ASIP was the failure of an F-111 in December1969. F-111A number 94 (SN 67-049) failed on 22 December 1969 as a result of a wing failure in the lower plate of the left wing pivot fitting. At the time of failure, the aircraft had approximately 100 hours of flight time. Catastrophic loss of this F-111 demonstrated the fatal defect in the "safe life" method. That is, the safe life method did not preclude designs that were intolerant to manufacturing and service-induced defects. Other losses (e.g., F-5, B-52, and T-38) and incidents of serious cracking (e.g., KC-135) during this period confirmed this shortcoming. These failures lead to a new approach for the protection of USAF aircraft safety, a damage tolerance approach. The approach selected by the USAF was damage tolerance. The concept of damage tolerance is discussed in detail in Section 2. The basis for the process is to assume the structure has a flaw, a sharp crack, that is the least upper bound of the expected flaw distribution. The operator makes inspections such that the crack does not reach the point of rapid propagation before it is detectable. The damage tolerance approach is in a state of continual improvement because research and development has lead to better methods in fracture mechanics methods and stress analysis over the last thirty years. The introduction of damage tolerance principles by the USAF in their structural inspection program in the early seventies virtually eliminated fatigue as a safety issue in their aircraft.

The USAF incorporated the damage tolerance approach in the ASIP, and in 1975, they published the process. This program, for a new acquisition, provides a series of time related tasks that will provide progressive risk reduction in the progression of the engineering and manufacturing development phase of procurement. The current version of the ASIP includes five separate tasks that cover all aspects of the development and support of an aircraft structure. For any given program, if the USAF does not plan to include a specific element, then they must establish the rationale and potential impact on the structural integrity of the weapon system for the exclusion. The main tasks of ASIP [2] are as follows:

I. Design Information
II. Design Analyses and Development Tests
III. Full-Scale Testing
IV. Force Management Data Package
V. Force Management

The original goals of ASIP were to (1) control structural failure in operational aircraft, (2) devise methods of accurately predicting service life, and (3) provide design and test approach that will avoid structural fatigue problems in future weapon systems.

The ASIP is also the standard by which the USAF can evaluate aging aircraft issues for structural components. For this purpose, the USAF normally emphasizes a subset of the elements of ASIP. For example, they extracted the appropriate elements of this program to perform the damage tolerance assessments (DTAs) during the seventies and eighties. The Air Force invested approximately one million man-hours in that effort to provide an inspection and modification program that greatly enhanced the safety of aging aircraft. Aging aircraft for many years have had a significant influence on the USAF research and development programs and have been a major driving influence on the elements of the ASIP.

Two of the main products of the ASIP process are development of the report on strength and operating restrictions and the development of the Force Structural Maintenance Plan (FSMP). If there is a need to change either of these documents because of flight beyond design usage that could introduce new critical areas, corrosion, WFD, or repairs, then the aircraft is said to be in a state of aging.

Experience with operational aircraft has shown they rarely fly according to their design spectrum of loads. Data from flight load recorders have typically shown there are considerable differences in usage severity among aircraft with the same designation. The USAF often finds the average aircraft usage is more severe than originally perceived early in the design process. This finding is made more significant by the fact the damage tolerance analysis may have not identified an area that would be a concern for aircraft with usage more severe than that assumed for design. Experience has shown the mass of an aircraft increases because of additional equipment or modification after an aircraft enters operational service. In addition, there are differences because there are changes in pilot techniques as they become more familiar with the aircraft, and mission changes because of new weapons and tactics. The aircraft-to-aircraft variability comes from several sources such as base to base variations in distance to test ranges and training. These experiences tend to degrade the capability of the full-scale durability test that consisted of two lifetimes of average usage to identify all the areas of the aircraft that could potentially cause a loss of safety. In most cases, an update of the DTA can account for any change needed in the inspection or modification program.

For the past forty years, the United States Air Force has used the USAF Aircraft Structural Integrity Program (ASIP) to maintain safe and economical operation of aging aircraft. This program has been supported over the years by USAF laboratory programs in the areas of fracture mechanics, corrosion prevention, flight loads, nondestructive evaluation, human factors, and maintenance and repair. These efforts provided the Air Force with the technology required to support the operational aircraft maintenance programs based on damage tolerance.

As indicated above, the USAF significantly changed this program because of the failure of an F-111 in 1969. This event ushered in the era of damage tolerance in the USAF [3]. The first assessments performed on the C-5A and the B-1A in 1971 and 1972 help derive the original DTA requirements for the USAF. These requirements were derived for monolithic (i.e., slow crack growth) structures. The failure of an F-4 wing on 23 January 1973 in a structural location the USAF considered fail-safe demonstrated to them that a structure could not be fail-safe without an inspection program. This failure strongly influenced the damage tolerance requirements as initially established first in MIL-A-83444 and subsequently in AFGS-87221A. The technology for the analysis of fail-safe designs has evolved slowly, primarily because of the need for extensive finite element programs supported by expensive test programs. The change to a damage tolerance approach prompted considerable research and development in the area of fracture mechanics. The then Air Force Flight Dynamics Laboratory was the focal point for much of this research. In the sixties and seventies, they developed much of the fracture technology that is still in use today. In addition, since the damage tolerance approach forced the engineer to better understand the stresses in the structure, finite element techniques emerged as the method of choice for the stress analysis. These capabilities permitted the USAF to perform a DTA of all the major weapon systems in the inventory in the seventies and eighties. This effort required over one million man-hours to complete and every major manufacturer was involved with this activity. Because of this activity, industry was able to develop the technology required for this type of analysis. This technology is also suitable for application to new aircraft developments. Consequently, the USAF was able to include damage tolerance requirements in the specification for new aircraft procurement.

After completion of the DTA on every major weapon system [4], the USAF laboratories continued research in other areas associated with aging aircraft. One of these was to make a better determination of the durability of aircraft. For this purpose, they sponsored research in the determination of initial crack distributions in aircraft structures. Much of this effort was concentrated on the interpretation of the cracks found in the teardown inspection of the F-16 wing after completion of the durability test. Another effort related to aging aircraft was the development of the procedure for the evaluating the probability of failure for a population of aging aircraft.

The need for nondestructive inspection technology to enable the damage tolerance driven inspections has been a major thrust of the Air Force for many years. Among these technology programs was a major effort to determine the probability of crack detection in an operational environment. Both the USAF and the FAA recognize the need for continuing the effort to quantify the capability of inspection techniques since this capability is critical to flight safety.

There are significant research and development efforts currently underway in the area of nondestructive evaluation of aging aircraft. NASA LaRC and several academic institutions including Iowa State University and Johns Hopkins University are doing much of this work. The USAF is working with these institutions and the FAA Technical Center to ensure these efforts meet the their requirements.

The USAF research and development program for aging aircraft has provided the technology base for safe and economic operation of military aircraft through the ASIP. As an indicator of this success, the failure rate for all systems designed to and/or maintained to the current policy is one aircraft lost due to structural reasons in more than ten million flight hours. This is significantly less than the overall rate of aircraft losses from all causes by two orders of magnitude. It has also, at times, given program managers a false sense about the remoteness of structural failures. This success, however, should not be used to indicate there is no need for continued research on the structure of aging aircraft. The return on the investment in this research is reduced cost and downtime with inevitable structural problems.

As indicated above, the materials in many aircraft were the result of the desire for improved performance with little attention given to the potential for corrosion and stress corrosion cracking damage. Further, at the time of manufacture of many of these aircraft, the focus on corrosion protection was not what it is today. Many of these early corrosion protection systems have broken down. In the open areas, the operator can readily renew them. There is, however, no easy way to renew the corrosion protection system in the numerous joints. Experience with modification and repair of aging aircraft has revealed that joints without proper protection experience significant damage that results in costly part replacements.

The corrosion concern is now becoming more acute in that the environmental protection laws have eliminated the use of some of the standard corrosion inhibitors. Another issue is that the nondestructive evaluation techniques are marginal. The standards for corrosion damage are so poorly defined that it is difficult to properly characterize the damage found. This deficiency creates a real problem in the future years cost projection for structural maintenance.

The Damage Tolerance Assessment (DTA) Process

The definition of damage tolerance is the following:

Damage tolerance is the attribute of a structure that permits it to retain it required residual strength for a period of unrepaired usage. It must be able to do this after it has sustained specified levels of fatigue, corrosion, accidental, or discrete source damage. Examples of such damage are (a) unstable propagation of fatigue cracks, (b) unstable propagation of initial or service induced damage, and/or (c) impact damage from a discrete source.

Figure 1 shows the steps in the DTA process. This description applies primarily to the process used by the USAF. The procedure used by commercial operators is quite similar.. The DTA is an integral part of the aging aircraft program for both military and commercial aircraft. The concept is simple. The flight time to the first inspection is based on the time required for the largest defect expected in a fleet of aircraft from manufacturing or in-service damage to grow to critical crack length. Subsequent inspections are based on flight time for the NDI detectable defect size to grow to critical crack length. A crack growth function illustrating this process is shown in Figure 2.

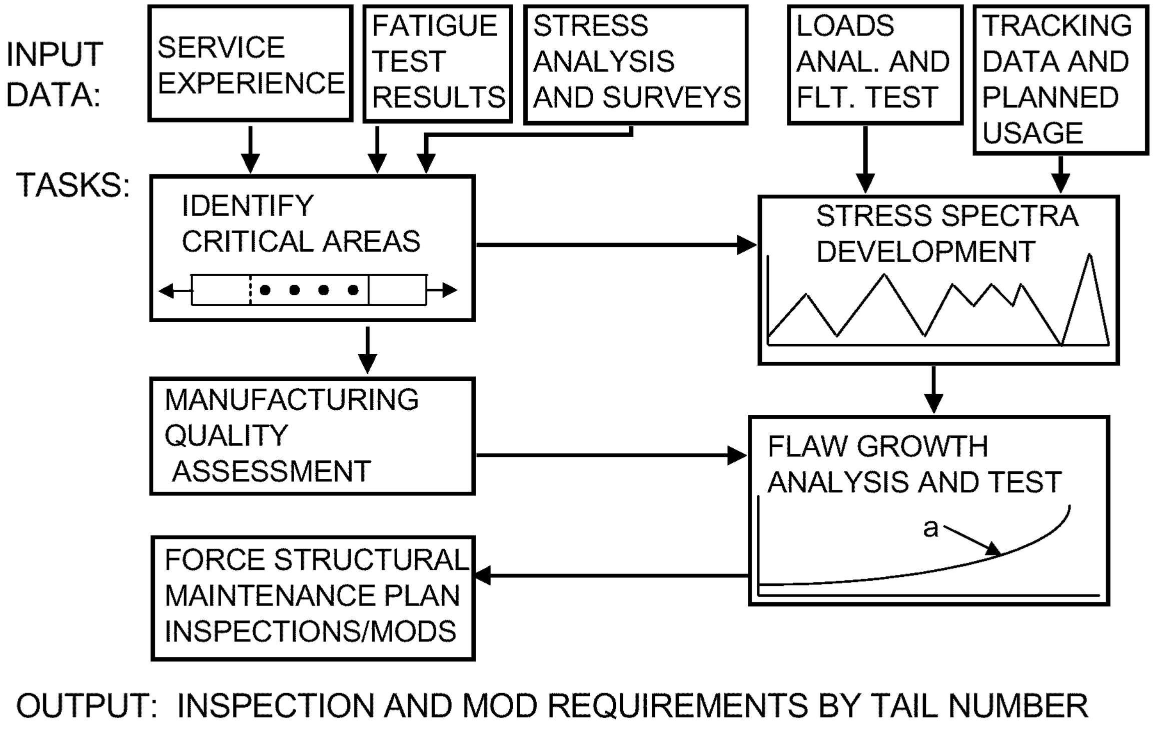
INPUT DATA:
SERVICE EXPERIENCE
FATIGUE TEST RESULTS
STRESS ANALYSIS AND SURVEYS
LOADS ANAL. AND FLT. TEST
TRACKING DATA AND PLANNED USAGE
TASKS:
IDENTIFY CRITICAL AREAS
STRESS SPECTRA DEVELOPMENT
MANUFACTURING QUALITY ASSESSMENT
FLAW GROWTH ANALYSIS AND TEST
a
FORCE STRUCTURAL MAINTENANCE PLAN INSPECTIONS/MODS
OUTPUT: INSPECTION AND MOD REQUIREMENTS BY TAIL NUMBER

Figure 1 The Damage Tolerance Process

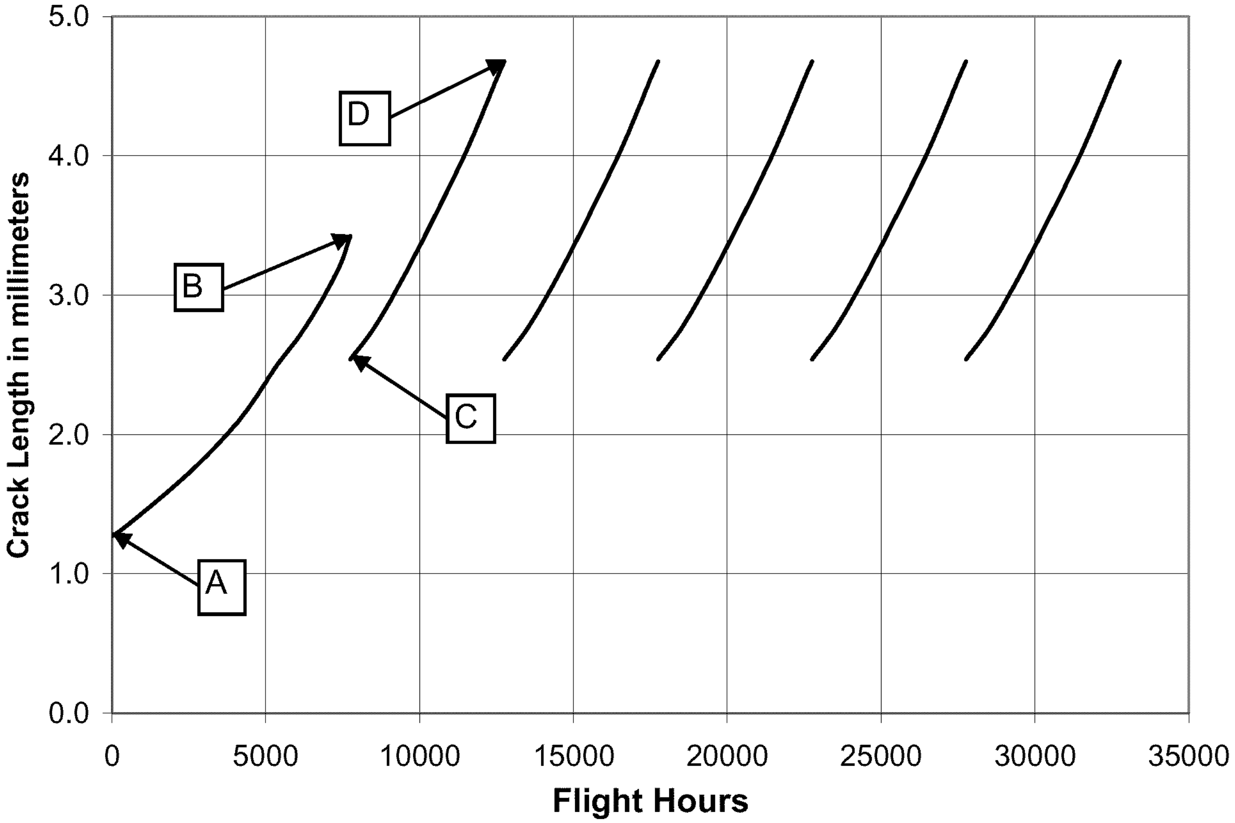

Figure 2 Damage Tolerance Crack Growth Function

The ordinate of the point A is the initial flaw assumed for the analysis. The abscissa of the point B is half the time needed for the initial flaw to grow to critical. The ordinate of the point C is the NDI detectable crack length. This crack is then grown to the point D whose abscissa is half the time required for crack to grow from B to critical crack length. The process is repeated until the inspections reveal an actual crack or the structure needs to modification for WFD.

The process evolved over a period of several years after the USAF applied it initially to the B-1A and the C-5A. Its successes include the F-4, an aircraft that did not have a requirement for life when the U.S. Navy procured it. The USAF purchased this aircraft in large quantities, and it became an essential ingredient of their fighter fleet. After a crash at Nellis Air Force Base in 1973 caused by fatigue, the USAF found themselves in a difficult situation. They initiated a recovery program that included a DTA and fatigue test conducted in their laboratory at Wright-Patterson Air Force Base. Because of this effort, the F-4 remained in operational service until the nineties without further incident. During the seventies and eighties, the USAF performed a DTA of every major weapon system in their inventory [17]. These successes motivated the USAF to apply this technology to engine structures with similar results. The discussion below describes the method used for damage tolerance with examples on how the USAF applied it.

The first task of the DTA is the identification of critical areas of the structure. A critical area is a location or part of the structure that could affect flight safety and may need maintenance in the form of an inspection or modification during the life of the aircraft. There are several techniques for identification of these areas. Actual cracking experience through service operations or durability testing is usually the most important consideration. Areas that have high predicted or measured stress and details that make them prone to cracking are, of course, prime candidates for the assessment. Another consideration in the selection of critical areas is its ease of inspection. In general, the analyst gives higher priority for selection on critical areas difficult to inspect. It has been extremely helpful to use the accrued knowledge of the original aircraft contractor in identifying potentially critical areas. On some of the assessments, preliminary estimates were made of the flaw growth in the candidate critical areas. When this of inspection. In general, the analyst gives higher priority for selection for areas that are difficult to inspect. It has been extremely helpful to use the accrued knowledge of the original information was available, it was much easier to make a decision on which of the candidate areas the analyst should subject to a final analysis. For small aircraft, the number of candidate areas generally was of the order of 40 to 70. The analyst would normally be able to screen these down to 10 to 30 for final analysis. For larger aircraft, the number of candidate areas generally was of the order of 60 to 150. The analyst would screen these down to 30 to 60 for final analysis.

The second task of the DTA is the development of the stress spectrum for each area identified for a final analysis. This is one of the more demanding aspects of the DTA process. The reason is there are significant changes in the rate of flaw growth due to relatively small changes in the cyclic stresses. To perform this task properly, generally three data items must be available to the analyst. First, he must have operational experience available in a usable form. This operational data must provide a basis for establishing a flight-by-flight sequence of points in the sky (i.e., altitude, weight, and aircraft motion parameters). This was usually available from multi-channel data on fighter or attack aircraft. For transport category aircraft, the USAF usually derived the sequence from flight log information supported by multi-channel data to define the maneuver and gust environment. In all cases except one, there was a sufficient database to derive the sequence. This exception was the A-7D, which was equipped with counting accelerometers only. Consequently, as a part of the A-7D DTA, an operational data base was derived from collecting 1,250 hours of multi-channel data from aircraft located at two bases. The second data item necessary for the derivation of the stress spectra is the set of equations needed to determine the external loads (i.e., shear, bending moment, and torsion) for a given point in the sky. For USAF aircraft, the manufacturer usually determined the external loads through analyses, wind tunnel testing and in-flight strain surveys. For all aircraft except the F-4, there was sufficient confidence in the existing data to perform this task. The USAF elected to perform a flight loads survey on this aircraft during the course of the DTA. This turned out to be very beneficial because the pre-existing data would have produced a pessimistic view of the maintenance burden for this aircraft. The final data item needed for the generation of the stress spectra is the external loads to stress transformation. For all of the aircraft studied, there were at least some experimentally derived stresses from previous static and durability tests. However, in all cases it was necessary to conduct additional stress analyses. The contractor performed these additional analyses typically using the finite element method. The scope of this finite element effort ranged from evaluating stresses at local details to finite element models of the complete airframe. The finite element effort varied significantly from aircraft to aircraft because of differences in the test database and the complexity of the critical details. Simplification of the stress spectra effort would have been possible if direct strain measurements had been available. In general, these data were not available. In a few cases, such as the C-5, this kind of information was available and was invaluable for determining the environment from maneuver, turbulence, and aerial refueling.

The techniques used in deriving the stress spectra for the assessments varied quite widely from aircraft to aircraft. Part of the reason for this difference was due to available database. For example, for the F-4, the data collected from the VGH recorder provided the number of occurrences of combinations of Mach number, load factor, and altitude in predetermined bands. Consequently, the assessment of areas of the aircraft sensitive to asymmetrical loading required data from other aircraft or from pilot interviews. For the F-15, however, the Signal Data Recorder provided a time history of both symmetrical and unsymmetrical parameters for use in developing the stress spectrum. The F-15 database more accurately accounted for the unsymmetrical loading. Moreover, it permitted a more realistic assessment of the minimum stress excursion that followed a maximum stress excursion. For the F-4, the conservative assumption had to be made that after a maximum stress there followed either a stress corresponding to one-g flight or a stress corresponding to less than one-g flight. The database on the F-15 enabled the analyst to remove this conservatism.

The USAF performed all of the fighter and attack aircraft assessments by reconstituting the individual flights from the databases except for the F-111. For this aircraft, the multi-channel recorder data was used directly to randomly generate a "block" of flights of approximately 500 hours. This is a very effective approach if one can be sure the selected flights are representative of the aircraft usage. For the F-111, they used the counter data for load factor as a guide for this selection. There was no attempt made to maintain the original order of the individual flights since previous studies for the F-4 and other aircraft showed sequence effects were insignificant if the flights were randomly selected.

As indicated earlier, the VGH recorder was the basis for the F-4 usage database. A sampling technique based on VGH recorder data provided the approach for the derivation of the stress exceedance function. In this method, the analyst computed the stress for a representative set of Mach number, load factor, weight, and altitude combinations. The surface derived from the representative points provided the means to determine the stress at the flight-measured points. Thus, the recorder data determined the stress exceedance function accounting for "all points in sky.” The USAF used a modification of this approach in the development of the stress exceedance function for the A-7D DTA. For the A-7D, the approach involved a regression equation to interpolate based on the stresses computed for a representative set of aircraft flight conditions.

The environmental data that augmented the flight log data for the large aircraft were extremely important. The USAF refers to these data, used in the ASIP, as the loads/environmental spectral survey (L/ESS) data. It provides the means to quantify the three dimensional nature of wing gust loads, the phasing of shear and bending moment, and the aerial refueling loads on the C-5. These data were also very helpful for evaluating the low-level turbulence on the B-52, C-141, and C-130. In many cases, such as the fire-fighting mission for the C-130, special mission maneuver data needed quantification. It is the intent in the derivation of the stress spectra to determine the "baseline usage" as an average usage for the force. For aircraft where there were significant usage changes during their life or there were possible changes in their future usage, the baseline usage reflected these changes. For some aircraft, such as the F-111, with different Mission Design Series (MDS), the USAF derived a separate baseline usage for each MDS. In addition to the baseline usage, there is a need to derive stress spectra that represent potential variations from the baseline. The testing of these variations develops confidence the procedure for tail number tracking by fracture mechanics methods is valid. For the older aircraft assessments, the usual procedure was to define a spectrum more severe than the baseline and a spectrum less severe than the baseline. Changing the baseline mission mix generally accomplished this.

For the larger tanker, transport, and bomber aircraft, the main source of data was the flight logs. In general, these logs had sufficient detail such that engineers could divide the usage among a relatively few missions (of the order of ten). Typically, the assessment had to include two or more distinct usage changes. For example, for the B-52Gs there were differences in usage prior to, during, and after their Southeast Asia operations. In addition, the USAF anticipated the usage of the aircraft in the future to be different from all the previous usage.

The third task of the DTA is to establish the initial flaw size for the fracture analysis. Because of their inherent stress concentration, fastener holes were predominant as candidates for critical areas of the airframe. The USAF noted there had not been a structural failure in the number of flights it takes for a 1.27 mm corner flaw in a fastener hole to grow to critical crack length. By 1975, they believed there was sufficient data to make the judgment that this size was sufficient to ensure aircraft safety. They derived this belief partially from teardown inspections of full-scale fatigue test aircraft, but primarily from observing operational aircraft such as the F-4, C-5A, and the KC-135. There has never been a rigorous substantiation of this belief. However, experience in subsequent years supports use of this size defect as being adequate to protect flight safety. The remaining task then was to determine the flaw size for holes that were cold worked or filled with an interference fit pin. The USAF determined this flaw size on ad hoc basis. In some cases, where there was a question of the adequacy of the installation of the interference fit pins, there was no reduction allowed. In other cases, where there was confidence the installation was proper, the USAF reduced the initial flaw size to 0.127 mm. The primary considerations in making this judgment were durability test performance and manufacturing procedures. For example, the C-141 durability test showed that the tapered fasteners did extend the life of that aircraft. However, there was some concern about the quality of the hand held drilling operations in some areas of the wing. Consequently, the DTA did not account for the benefit of the tapered pins. However, machines with controlled feed and speed drilled many of the wing fastener holes. For these holes, the USAF made a decision to reduce the initial flaw by approximately a factor of two.

The identification of the stress spectra for each critical area and the initial flaw permits the initiation of the task of establishing operational limits. This combined analysis and test effort uses the disciplines of fracture mechanics to find the safety limit for each critical area. The fracture mechanics technology has improved significantly from the early 1970 period. However, even with the analytical capability available today, the process would be meaningless without test substantiation.

There are two main reasons for fracture testing. First is analysis verification. The aim is to accomplish this with the least specimen complexity possible in order to isolate the local detail (e.g., a fastener hole) and evaluate the spectrum retardation. The specimens used for this purpose were generally dog bone specimens with the proper material, thickness, size of fastener, and load transfer. The second reason for testing is to establish high and low side truncation levels. The low side truncation is primarily an economic consideration. The object is to eliminate as many cycles as possible with a small stress range without significant change in the crack growth. The removal of the high stresses (or clipping) eliminates those cycles that make a crack grow slower and retain those which make a crack grow faster.

In the early 1970 period, there was a belief that crack growth was quite sensitive to the loading sequence within a flight. Therefore, as part of the DTA, the USAF required tests to evaluate these effects. It was learned that if they simulated loading on a flight-by-flight basis, then the ordering of loads within a flight was of secondary importance. Consequently, they discontinued this type testing. For almost all of the aircraft, the fact that the critical crack sizes were sufficiently small such that there was little if any redistribution of stresses during crack growth simplified the fracture analysis. Further, the primary structural issue was crack growth from a fastener hole or an open hole. Therefore, the analyst needed to concern himself with the part through flaw in mode I cracking from both filled and unfilled holes, load transfer on the fasteners, and retardation effects. For simulating the retardation effects, the analysts generally used the Wheeler model, the Willenborg, the modified Willenborg, or some form of a contact stress model. The T-38 analysis used the Vroman model for part of its DTA. This model was not used for any other assessment.

Analysts learned that proper counting of the stress cycles in the spectrum was essential for obtaining accuracy. The so-called rain flow procedure is now commonly accepted as an adequate procedure for counting the stress cycles in the spectrum [30].

After the verification of the crack growth analytical model through coupon and in some cases, component testing, the effects of the chemical environment entered the analysis process. For the early assessments, the tendency was to take a conservative view of the environment. That is, the USAF required the selection of an environment more aggressive with respect to crack growth than actually expected. This position was relaxed in the late 1970s and they placed emphasis on selecting a realistic environment. Constant amplitude crack growth tests performed in the desired environment provide the basis for the quantification of these effects. The crack growth analysis includes the environmental effect in the data used for the crack growth rates. The procedure is subject to criticism because it may not accurately account for the effects of cyclic frequency and load interaction effects with the environment. There is no indication from the inspections performed on operational aircraft the error is significant.

The crack growth analysis plays a dominant role in damage tolerance approach. The tool must be usable for different chemical as well as loading environments. In other words, it is the mechanism for tracking the crack growth on each tail number in the force and thereby ensuring aircraft safety. Therefore, it is extremely important to validate the analysis for the expected range of service operations. After the analyst establishes the safety limits for all the critical areas of the structure, the development of the Force Structural Maintenance Plan (FSMP) can proceed. The FSMP provides the how, when, and where for structural inspections or the when and where for modifications. In many cases, it was found, based on either economic or safety considerations, that modifications were preferable to continued inspections. This situation existed for the C-5, T-38, F-4, and KC-135, for example. Of course, the DTA process should include the modifications.

One of the more important tasks in the damage tolerance approach was to establish the NDI capability. This was done with the help of the NDI experts from the now Air Force Research Laboratory (AFRL) Materials Directorate, the appropriate Air Logistics Center (ALC), and the contractor. In some cases, such as the EF-111, they conducted an NDI reliability program to determine the flaw size corresponding to 90 percent probability of detection with 95 percent confidence. However, these cases were in the minority and, consequently, the USAF based most of the NDI detection capability on judgment. When possible, they avoided inspections that involved removal of fasteners. In addition, the USAF rejected the concept of sampling inspections rather than inspecting 100 percent of the force.

The FSMP covered the period of the planned operational usage of the aircraft. Thus, the FSMP permitted the ASIP manager at the ALC to determine the out years maintenance cost. The accuracy of these costs was suitable for budgetary estimates. The accuracy for any given tail number is; of course, dependent on how closely that aircraft flies to the baseline.

USAF structural engineers have long recognized the need for tail number tracking of aircraft. This is evident from the emphasis given to it in the 1959 version of ASIP. The only significant change from the original version is the tracking process is for crack growth rather than fatigue damage. The USAF developed the first tracking program based on fracture for the F-4 during its DTA. Now all aircraft that have had a DTA have a tracking program based on fracture mechanics. For many of these aircraft, the ASIP manager has the computer programs to provide an immediate view of the maintenance status of his aircraft. This provides him with both near and far term planning and decision making capability. It provides him with the capability to determine the consequences of a mission change. There is also a need for commercial aircraft to have a periodic reassessment of their usage. The availability today of excellent digital recording devices has made this task considerably more manageable than in the past.

The damage tolerance approach has led to a greatly improved understanding of aircraft structures and their performance. It is the foundation for maintaining flight safety in aging aircraft. It has also led to a greater recognition that additional research and development in the areas of materials, structures, and nondestructive evaluation were not only needed, but could further increase the reliability of systems. Consequently, over the last several years, many programs have focused themselves on increasing the knowledge base available to enable longer lives and more reliability from airframes and engines. Overall, the damage tolerance experience has been good. The criticism, which is rare, has come from people who believe the approach is too conservative when they perform an inspection, and find no cracks. On the other hand, the DTA process has correctly directed inspections to areas that full-scale testing did not indicate they were critical. Figure 3 shows the DTAs performed by the USAF during the seventies and eighties.

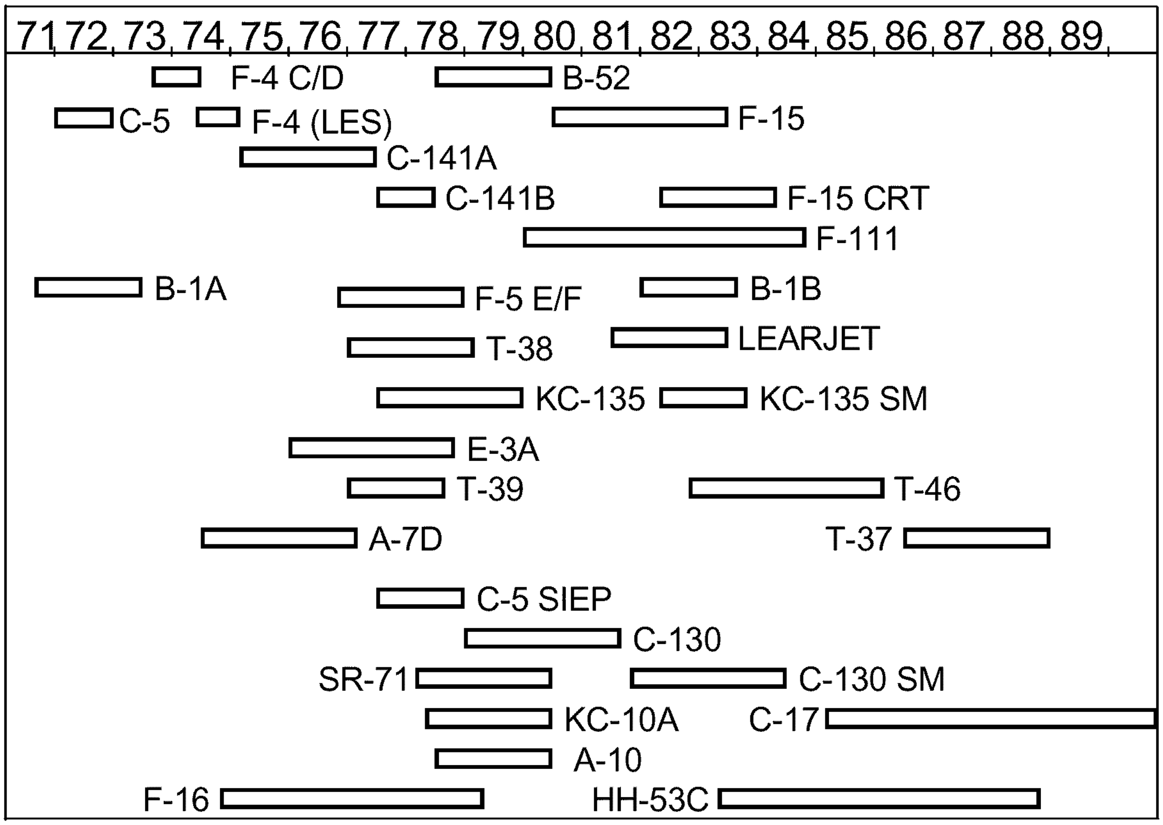

Figure 3 Damage Tolerance Experience in the USAF

Conclusions

Operational aircraft failures from fatigue in the fifties and sixties motivated a fundamental change in the approach for ensuring safety of flight for aircraft. In the seventies, many certification authorities endorsed the damage tolerance process for design and maintenance of safety critical structure. The process uses stress analyses, loads analyses, and fracture mechanics to determine inspection intervals or modification times to the in-service maintenance program. This disciplined process has proven to be successful in preventing structural failures from fatigue.

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